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A 4-FLAP DESIGN WITH GRAPHITE AND BORON COMPOSITES

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A-4 FLAP DESIGN WITH GRAPHITE AND BORON COMPOSITES

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Abstract

The design and development of an A-4 landing flap is described. The flap was initially selected as a suitable component for obtaining flight experience with boron filaments. The existing production aluminum flap was redesigned utilizing boron skins supported by a full depth honeycomb core, while retaining some of the basic component aluminum parts for interchangeability reasons. A second flap design, utilizing graphite as the composite reinforcement and developed in accordance with advanced structural element concepts such as molded graphite parts to replace aluminum rib and hinge fittings was developed. The choice of skin layup and core geometry is explained. It is shown to depend not only on the strength and stability of the skin but also on the temperature and pressure conditions that exist during the cure and secondary bonding. Emphasis is placed on the problems encountered during the design and development phases rather than on those which came to light during the final detailed analysis.

Introduction

Early in 1967 it was decided that composite technology was sufficiently advanced for research work to proceed on a design, applicable to composites, of a significant flightworthy component. Consequently, Douglas initiated an Independent Research and Development program that would use boron filaments in an epoxy matrix as the primary composite material. After a survey of candidate components, the A-4 Landing Flap was selected as having the right amount of complexity without being prohibitively difficult. The skins of the flap were thought to be a good choice for composite materials since they were designed with torsional shear as the primary loading condition. Although slightly twisted, they could be made initially by flat layup techniques. The need for automatic fabrication methods was anticipated and a tape layup machine was designed and developed to assist in the manufacture of the skins.

A study to determine the effect of various materials on the total weight of a projected aircraft showed graphite to be a promising material. In order to gain experience in the use of graphite filaments, this material has been made the subject of a further Independent Research and Development program. As part of this work, the flap is being redesigned to utilize graphite materials. Changes to the design result from lessons learned from the boron flap and from the basic differences in the two filamentary materials. Investigations have been made into the advanced technology of molded graphite fittings and into the use of a graphite honeycomb core.

The Production Aluminum Flap

The existing metal flap is approximately six feet span and two feet chord and has a constant triangular section about two inches deep at the leading edge, Figure 1. The flap is mounted on the wing structure by means of a piano-hinge along the lower surface leading edge. The actuator attachment is at a forged aluminum rib fitting at the inboard end. An access door is provided through the upper skin to facilitate the fastening of the actuator attachment bolt. A landing gear fairing structure is attached to the lower skin by eight bolts.

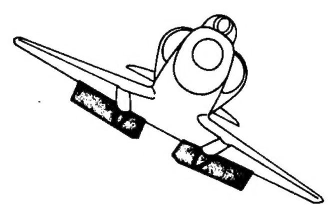

FIGURE 1. A-4 AIRCRAFT WITH LANDING FLAPS SHOWN SHADED

The flap skins are made from 0.032-thick, 7075-T6 aluminum and are supported on closely spaced ribs, the pitches of which are about three inches at the inboard end and six inches at the outboard end. The trailing edge and piano-hinge members are machined from aluminum extrusions. The remainder of the structure consists of sheet metal parts and, apart from a few bolts, the assembly of the flap is entirely by riveting.

Design Loads

The loads report for the metal flap shows that the primary loading condition occurs during the landing approach when the flaps are down. The air loading is assumed to be uniform along the span and builds up a hinge moment from zero at the outboard end to 17,750 inch-pounds at the inboard end. Since the actuator is not at the inboard extremity of the flap, the hinge moment associated with the design of the torsion box is reduced to 17,400 inch-pounds. A simple torsion calculation on a single-cell box yields a shear loading of 360 pounds per inch. This takes no account of the multi-cell nature of a honeycomb core, the distribution of hinge reaction along the span, nor of stress concentrations adjacent to the inboard rib and door cutout. However, this value was satisfactory for initial design purposes and it was subsequently found to be quite close to the maximum value obtained from a full stress check.

A secondary design condition occurs when the flap is up, during negative maneuvers at transonic speeds. This condition puts chordwise bending between the hinge and the trailing edge, which is insignificant except for the fact that the skin layup pattern is optimized for carrying torsional shear. Further complications arise because the flap structure is twisted each time it is pulled up against the upper wing structure by the actuator. This twist is built into the flap during assembly and is intended to prevent "buzz" during flight. However, since the flap is retracted twice in every flight, this twisting becomes an important part of the fatigue cycle. The built-in twist of the metal flap was one inch, but it was estimated that half this amount would be more suitable for the boron flap and this was subsequently built into the bonding jig.

An ultimate factor of 1.5 was applied to all limit loads and it was decided that, for the composite components, neither buckling nor initial failure would be permissible before ultimate load. The post-buckling behavior was not thought to be satisfactory and it was argued that initial failures within the laminate would cause a permanent degradation of the material. This is in marked contrast to the permanent deformation of metallic materials which is permitted between limit and ultimate loads. Hence, limit load has no real significance for the composite parts of the structure, as they are all designed to meet ultimate load conditions. This stipulation imposed no penalty on the flap skins where the minimum practical thickness was used.

Designing with Boron

The basic material is a 0.0040-diameter filament formed by vacuum-depositing boron onto a 0.0005-diameter tungsten substrate. It has a tensile modulus in the region of 60×10^6 psi, and its remarkable stiffness precludes bending it to a radius of curvature of less than three-tenths of an inch. Below this radius, the maximum tensile strength of about 400,000 psi is exceeded. In practice, the minimum design radius is considerably more than this, and even then, full account has to be taken of the residual bending stresses. For the flap, the boron was restricted in use to flat skin panels only, and even the bending caused by joggling one filament over another was avoided. For this reason, the filaments are arranged in layers and are collimated to lie parallel within each layer. The collimation takes place during the impregnation process in which each filament is coated with resin and arranged into a tape. This stage was completed by Narmco who supplied a three-inch-wide tape impregnated with their 5505 epoxy resin system. The tape has a glass scrim cloth carrier and a paper backing to prevent it sticking together when wound on a drum. The quality of the tape is controlled by a Douglas Material Specification so that the resulting laminates are of an acceptable standard.

The laminates are prepared by building up one layer of filaments at a time. When tapes are laid next to one another, both gaps and overlaps at the edges of the tapes are undesirable, since they lead to loss of strength. It is a particularly difficult problem with an automatic layup machine to obtain the desired standard of edge alignment. The filaments in one layer can be placed in any desired direction and the complete arrangement of layer directions is usually referred to as the layup pattern. This pattern is selected to give the required strength in all directions and, in the case of thin skins, the required stability of the panels.

It is important that the layer directions be arranged symmetrically about the mid-plane of the laminate in order to achieve thermal balance. This is necessary because each layer has markedly different thermal coefficients of expansion in the longitudinal and transverse directions. The coefficients for boron and the resin are, respectively, about 3 and 40×10^{-6} per degree Fahrenheit so that, during the cooling stage of the cure cycle, the contractions are very different in the two directions. Unless this effect in one layer is balanced by a corresponding effect on the other side of the mid-plane, considerable warping may take place. The overall expansion of the whole laminate must be considered when joining to another material. This other material could be a metal component or even another laminate with a different pattern. As an example, the coefficient of expansion has been derived for a laminate having a $\pm\phi$ pattern, Figure 2. A comparison is made with aluminum, titanium, and steel. It can be seen that a zero degree unidirectional laminate is closely matched by titanium but that a ±45° pattern would be better combined with either steel or aluminum. Obviously, care is required to prevent warping when dissimilar materials are joined by a method involving temperature.

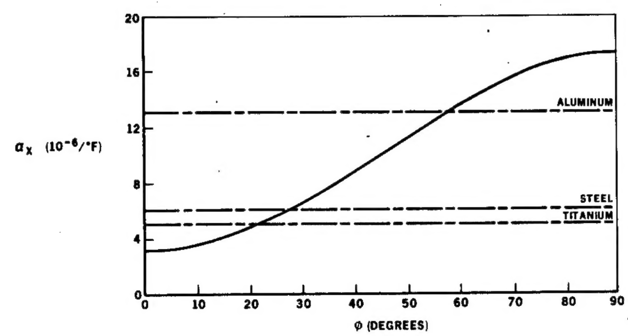

FIGURE 2. COEFFICIENT OF THERMAL EXPANSION, BORON LAMINATE WITH ± ϕ PATTERN

The joining method chosen for the flap was by adhesive bonding. This is a logical choice since the composite materials themselves rely entirely on adhesion for their integrity. Adhesive bonding does not enjoy the same confidence level that riveted and bolted constructions enjoy and yet a bonded joint can be not only strong and light but can offer a considerable improvement in fatigue life. This lack of confidence is not really justified with modern adhesives, which now have an extensive background of development and operational experience. Tack rivets, once a popular means to prevent peeling, were not used on the flap since they were considered to be of little or no value, and possibly even detrimental because of the presence of the holes. The joints were mainly of a lap shear type, for which an adhesive requires considerable ductility to accommodate the deformations due to the strain incompatibility of the adherends. The 350° F cure adhesives were rejected because some of the aluminum parts are adversely affected at

these temperatures. The selection eventually fell on Narmco 252, a film adhesive with a 250° F cure. This reduced temperature gave the added advantage that thermal stresses and distortions would be alleviated. Even at this temperature, problems with dimensional changes were encountered and the design procedures for the graphite flap have been modified as a result. The possibility of a good room-temperature adhesive is now being investigated. Use of such an adhesive would not only eliminate the thermal design problems but would also greatly ease fabrication.

Basic Boron Properties

A program of testing to find basic property data was initiated and, at the same time, much analytical work was being done to predict laminate strengths. Since much of this work was incomplete, a survey was made into all the test results that could be found from available literature and this work was subsequently published in compendium form. An estimate was then made of the strengths that could be expected using the best techniques available at that time. To use the analytical program that gives laminate properties it is necessary to input the strengths and stiffnesses of a single layer. These properties are often referred to as "monolayer" values, and are listed for boron in Table I, together with those for graphite and comparative values for 7075-T6 aluminum. It should be noted that the values given for boron and graphite are for initial design only and represent average values likely to be achieved. No statistical analysis was available at that time to show the variability of results that could be expected. A Douglas Process Specification was written to ensure uniform standards of fabrication and a high confidence level. Variability is particularly difficult to control in the transverse direction in which, for example, the presence of voids in the matrix has a major effect on strength, Figure 3.

Initially, residual thermal effects were also included in the computer program, for which the coefficients of thermal expansion of boron and the matrix had to be specified. However, it was found that better agreement was obtained when the thermal effects were omitted, suggesting that the residual strains were not significant. This may have been due to the fact that the visco-elastic nature of the matrix eventually relieves these strains.

The effect of inclining the filaments with respect to the loading axis was also investigated and empirical curves for laminates with $\pm\phi$ patterns were produced for both tension and com-

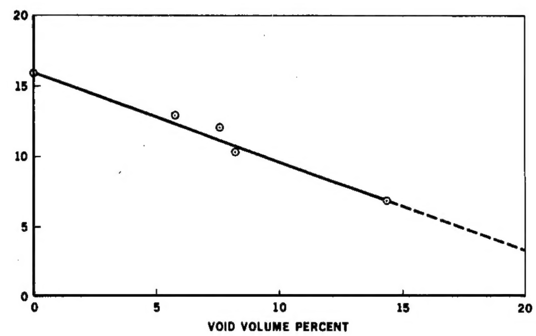

FIGURE 3. BORON TRANSVERSE FLEXURAL STRENGTH VS. VOID CONTENT

TABLE I
SUMMARY OF MATERIAL PROPERTIES
DESIGN MONOLAYER VALUES

MATERIAL PROPERTY		BORON	GRAPHITE	7075 T-6
LONGITUDINAL MODULUS	E_L (10^6 PSI)	32	25	10.3
TRANSVERSE MODULUS	E_T (10^6 PSI)	3.2	1.0	10.3
SHEAR MODULUS	G_{LT} (10^6 PSI)	1.1	0.6	3.9
POISSON'S RATIO	μ_{LT}	0.25	0.30	0.33
POISSON'S RATIO	μ_{TL}	0.025	0.012	0.33
LONGITUDINAL TENSILE STRENGTH	σ_{L_t} (PSI)	200,000	100,000	76,000
LONGITUDINAL COMPRESSION STRENGTH	σ_{L_c} (PSI)	250,000	80,000	67,000
TRANSVERSE TENSILE STRENGTH	σ_{T_t} (PSI)	10,000	5,000	76,000
TRANSVERSE COMPRESSION STRENGTH	σ_{T_c} (PSI)	24,000	15,000	70,000
SHEAR STRENGTH	τ_{LT} (PSI)	12,000	5,000	46,000
LONGITUDINAL EXPANSION COEFFICIENT	α_L (10^{-6}/°F)	3	0	13
TRANSVERSE EXPANSION COEFFICIENT	α_T (10^{-6}/°F)	17	17	13
DENSITY	ρ (PCI)	0.072	0.052	0.101

pression loads. These curves, being based on test results, were preferred to the results that were yielded by the analysis, although the agreement was reasonably close. The curves were later used for a rapid assessment of laminate strength by superimposing the strengths of individual layers. While not having the accuracy of the computer analysis method, it was found to be an extremely useful design tool for quickly converging on a suitable pattern. An attempt was made to adapt the method for combined loading but the results were not always very good. For shear, in particular, the predicted strengths were much higher than those actually achieved. Shear strength is also the least reliable quantity derived from computer analysis and furthermore, the various test methods used to find it are often the cause of much controversy.

Test Specimens

It was found that a cylindrical torsion specimen gave good agreement with the theoretical shear modulus, but that a picture frame shear sandwich specimen was adequate for predicting the strength of the skin when supported on a sandwich core. One objection to the cylindrical type of specimen is that the quality of the laminate does not equal that of a flat plate. This is because the uncured layers are thicker than their required finished dimensions and therefore, the outer layers have an excess of diameter which has to be lost during the cure. The layers are not always capable of accommodating the resulting reduction in circumference without a certain amount of wrinkling and loss of quality. This is more evident with the stiff boron filaments than with either glass or graphite and relates more to circumferential than to longitudinal layers. Since this effect accumulates with thickness, it is less serious when the wall of the cylinder is thin.

The picture-frame shear method is usually criticized because of the restraint imposed at the edges by the frame and because it is difficult to obtain uniform shear conditions within the test area. However, this method was selected for two reasons, the first being that it seemed reasonable to simulate a flat plate in shear, rather than to use a method that introduced curvature in the laminate or that derived shear properties in some indirect way. Secondly, it was clear that the laminate was going to be very thin and would require continuous support to prevent buckling. For these reasons a small sandwich cruciform specimen was designed to match an existing test frame, Figure 4. The results of these tests are discussed later.

Even the results of simple tensile tests require some interpretation. There appears to be a width effect, so that a narrow specimen may not accurately predict the strength of a wide sheet. This could be due to edge effects, possibly due to shear lag within the specimen. Furthermore, when angled layers have filaments that run out at the edges of the specimens the full strength of the filaments cannot be utilized. The load has to be transferred across an interfilament boundary and the strength of the layer depends on the principal tensile stress in the matrix. As the strength is limited by the tensile strength of the resin, it is possible to plot the strength of a unidirectional laminate against the angle of the filaments, Figure 5. This is compared with the test results for

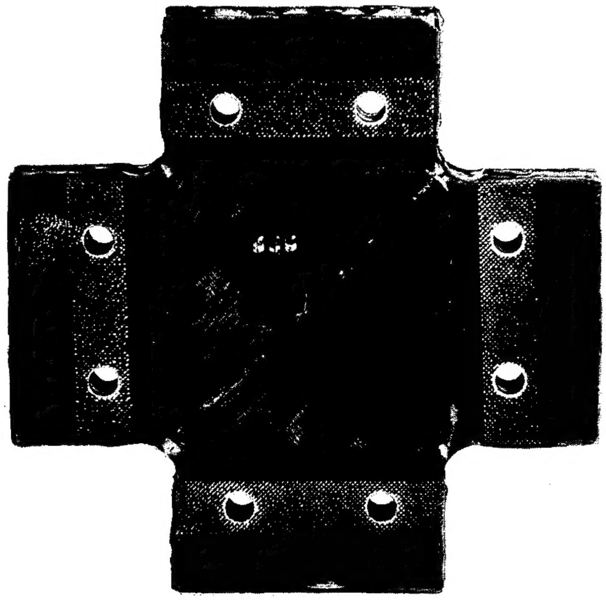

FIGURE 4. EDGEWISE SHEAR SPECIMEN

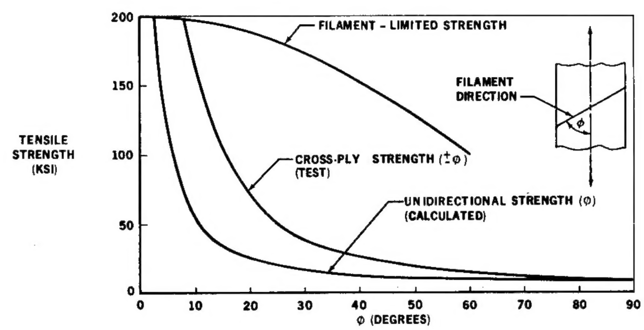

FIGURE 5. TENSILE STRENGTH OF BORON LAMINATES

crossed layers ($\pm\phi$) and it can be seen that these are stronger than the unidirectional layers. Since each individual layer has the same stress conditions, the increase in strength can be regarded as a synergistic effect due to crossing the filaments. Obviously, a shear failure along a filament line in one layer cannot have a corresponding failure line in an adjacent cross-layer without shearing the filaments in that layer. Boron laminates tend to fail along a common line so that there is some shearing of the filaments. Glass is more likely to fail along filament lines in individual layers with consequent delamination between the layers and graphite may behave in a similar way. If members are placed at the edges of a tension panel such that the filament loads could be transferred to these members, then the full filament strength might be developed, if transverse failures within the laminate do not occur first. The resulting limiting strength is shown, for comparison, in Figure 6. This discussion has important implications when designing actual structural components in which edge members are usually present.

Interlaminar shear strength is found by the usual short beam test in which failure occurs between the layers rather than by bending of the beam. For a unidirectional zero degree beam,

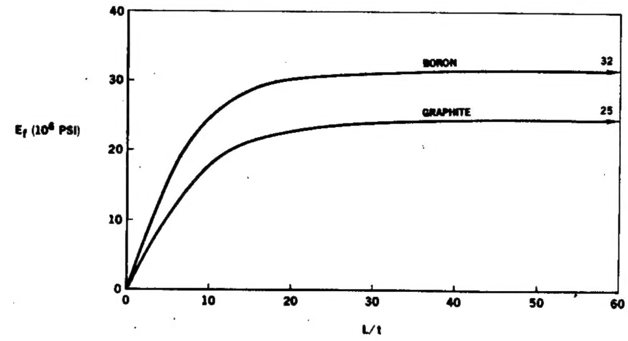

FIGURE 6. APPARENT FLEXURAL MODULUS, 0° LAMINATES

the interlaminar strength is related closely to the monolayer in-plane shear and transverse tension strengths. These, in turn, depend on the tensile strength of the matrix. When the layers are angled within the laminate there is a considerable drop in inter-laminar strength, a ±45° laminate having only about one-third of the strength of a 0° laminate. When different layer angles are used, a full shear distribution within the laminate must be calculated, since the usual isotropic distribution no longer applies. The interlaminar shear property is important for many design conditions and one of these is in bonded joints, where failure often occurs within the laminate rather than in the adhesive or at the interface.

The low shear stiffness between the layers compared with the high stiffness of the filaments along their length means that the shear deflections of beams cannot always be ignored. For example, in the flexural test on a 0° beam, commonly used as an acceptance test, the apparent modulus is affected by the length/thickness ratio, Figure 6.

Similarly, a sandwich beam specimen, which was used to study the behavior of the skins in tension and compression, failed in shear rather than in bending, Figure 7. This was due to the fact that the core used was a very light aluminum honeycomb and also because the loads were placed too close to the supports so that the shear was unnecessarily high.

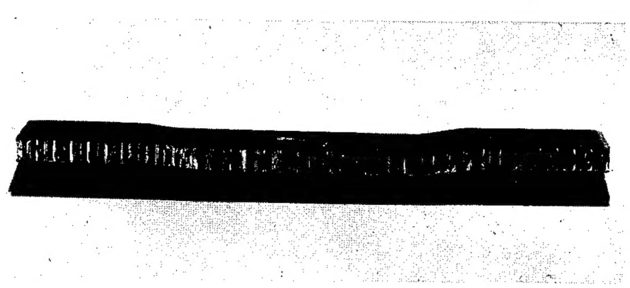

FIGURE 7. SANDWICH FLEXURAL SPECIMEN FAILURE

One very relevant test is the flatwise adhesion sandwich test, Figure 8. This is particularly important, in that it demonstrates the tensile strength between the skin and the core, a property that is required to find the buckling strength of the skin. In actual fact, all the specimens failed in the core itself, showing that the bond was more than adequate.

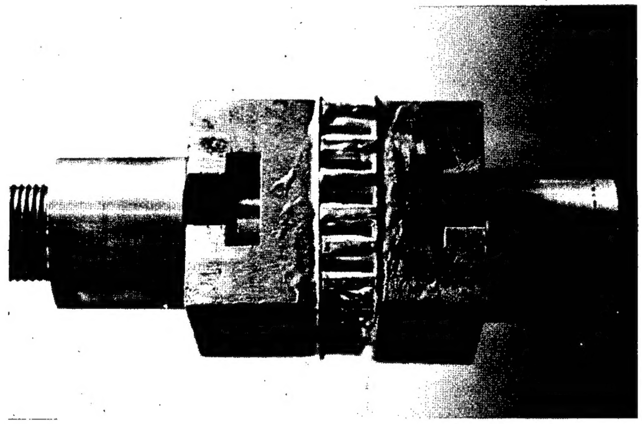

FIGURE 8. FLATWISE ADHESION SANDWICH TEST

Sandwich Design

It was immediately obvious that the loading was not sufficiently high to require a thick skin, but it was not clear just how thin the skin should be. One layer was not enough because the shear strength of a monolayer is so low, depending as it does on the resin strength between adjacent filaments. Two layers at ±45° would appear to be an efficient shear structure but this was rejected because of the thermal imbalance of such an arrangement. Three layers arranged (+45/-45/+45) constitute a structure that is in thermal balance but is not balanced in shear. As far as strength is concerned, this can be offset to some extent by putting two layers in the weaker tension direction and one in compression. The main objection to three layers is that it is difficult to analyze, as some secondary coupling terms in the elasticity equations cannot be omitted. Four layers is the minimum that can achieve both thermal and shear balance, by having a pattern of (+45/-45/-45/+45). This arrangement gives a very different moment of inertia with respect to the two 45° axes and at first it was thought that this would adversely affect panel buckling. Further investigation showed that a panel held at all four edges buckles with curvature in two directions, thus making the effective I value a function of properties in both of these directions. This was confirmed by testing sandwich panels in shear, with the outer layers consecutively in tension and compression. When sandwich specimens with four-ply skins supported on a 0.500-thick aluminum honeycomb core were tested, the failures occurred at an average shear stress of about 45,000 psi. Similar specimens with three-ply skins yielded slightly higher and lower stresses when the outer layers were in tension and compression, respectively. The closeness of failure stresses for the two thicknesses seemed to indicate that failure occurred at the ultimate stress and was not due to any buckling phenomena. At the same time, the stresses were lower than expected and there was some evidence that the initial failure was in the compression direction, followed by a complete tension failure (see Figure 4).

An investigation into the stability of an orthotropic facing material on a sandwich core showed that there was a lack of good design information in this field. The latest proposed amendments to Mil-Hdbk-23 became available and these were

compared with other published literature.(1) The problem of a full depth core, supporting skins loaded by equal and opposite shears was not dealt with, but it could be assumed that each skin behaved independently of the other. The intracell buckling and shear crimping failure modes were found not to be critical, but skin wrinkling had to be checked. The empirical equation of Reference (1) was selected, since it applied specifically to a skin supported on a honeycomb core. This gives wrinkling stress as:

$$F_w = \frac{0.82\left(\frac{E_c t}{E' t_c}\right)^{1/2} E'}{1 + 0.64K}$$

where

E_c = core elastic modulus in a direction normal to the facings

t_c = core thickness

t = facing thickness

E' = effective facing modulus in the direction of the applied load

$K = \frac{\delta E_c}{t_c F_c}$

δ = initial deflection of facing waviness

F_c = flatwise sandwich strength (the lesser of flatwise core compression or sandwich flatwise tension)

This expression specifically applied to a single compression loading but it was assumed to be equally applicable to shear loading. For this reason, the skin was treated as a (0/90°) laminate with respect to the direction of the compression load. Since the filaments are almost elastic along their lengths, no plasticity effects were included. The effective modulus then becomes:

$$E' = \sqrt{E_o E_{90}}$$

For a given core, E_c and F_c are known from manufacturers' data. If the skin thickness is specified and the core thickness is defined by the geometry, only δ is unknown. This quantity was evaluated in terms of the wavelength of the initial deflection. The critical wavelength is the most dangerous and it is defined by Yusuff as:(2)

$$\lambda = 2.614t\left[E_f^2/\left(E_c G_c\right)\right]^{1/6}$$

where

$E_f = E'$

The initial deflection is then given the value: $\delta = \lambda/500$.

The quantity G_c is the shear modulus of the core in the xy plane, where x is in the direction of the load and y is perpendicular to the faces. The core ribbon direction for the flap was chordwise, since it was ordered in HOBE (unexpanded) form, and the value of G_c was taken to be midway between the spanwise and chordwise values. The wrinkling stress found by these means was more than the ultimate strength value, so that this mode of failure did not appear to be critical. The four-ply skin was selected to avoid the uncertainties associated with the three-ply design. This was combined with a Hexcel aluminum 3/16-5052-0.001N honeycomb core, which was readily available and easily met the skin support requirements.

The Boron Flap

The main consideration with the boron flap was to make it completely interchangeable with the metal flap, Figure 9. To simplify this, the decision was made to retain the production piano-hinge, the inboard rib fitting, and some of the associated sheet metal details. The use of boron was to be limited to the skins, with a separate laminate being made for the upper and lower surfaces. The skins had to be well supported against buckling and this was achieved by a full depth honeycomb core, Figure 10. This core had to be cut away under the access door to the actuator bolt and a surrounding structure to this cut-out was made with a fiberglass "dish" molding, Figure 11. The vertical webs of this dish pick up the shear loads from the edges of the core and transfer them to the rib fitting. The top and bottom flanges make a shear connection to the skins. In the area of the core cut-out, the bottom skin requires stiffening against buckling and an aluminum doubler is used in this region. This doubler passes under the flanges of the dish and extends under the rib fitting and the piano-hinge, helping to introduce the loads more gradually into

FIGURE 9. BORON FLAP

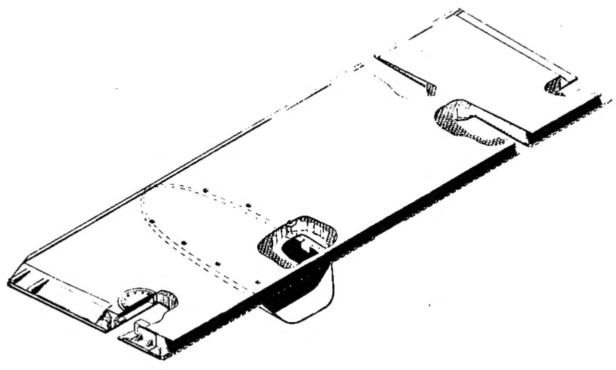

FIGURE 10. STRUCTURAL ARRANGEMENT

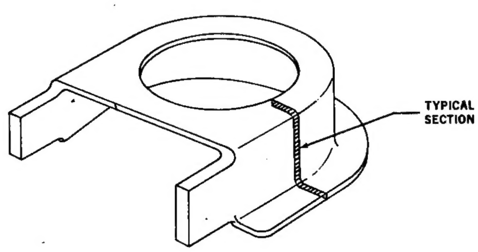

FIGURE 11. FIBERGLASS DISH MOLDING

these members. It also helps to reduce the peak stresses in the forward, inboard corner of the skin. Not only were the shear loadings highest in this corner, but it was thought that the rib might produce a warping constraint that would superimpose spanwise loadings as well. Subsequently, it was found during test that, as expected, failure did originate from this corner.

A similar aluminum doubler was fitted to the upper skin. This skin had a 2-7/8-inch-diameter hole cut in it and an aluminum reinforcing ring was added as an edge member. This ring was designed to transfer the skin loads around the hole, assuming the door to be non-load-carrying. The production aluminum door was retained since it was thought that it should be interchangeable with those on any other flap. For this reason, the existing ring of 13 countersunk bolts was used for its attachment and the bolts were screwed into nutplates fixed to the fiberglass dish. Initially, it was intended to rivet the nutplates to the dish, but not through the boron skin; the problem of drilling very small holes in boron would thereby have been avoided. However, it was later decided to bond the nutplates in position, a simpler solution. It still remains to be seen whether these bonds will withstand rough handling in service.

Fiberglass parts are also used to close the outboard end and the trailing edge of the flap. An outboard closing rib was molded from woven glass cloth and was basically only 0.030 thick. The web of this rib was inclined at 45°, a feature that was later to give trouble during fabrication. It was not appreciated at the time that, although the honeycomb core could withstand the bonding pressure in a direction normal to the skins, it had no capability to react the component of pressure parallel to the skins. This caused the web to deflect inward and crush the core. Later this problem was overcome by blanking off the pressure in this area. Midway along this web, a recess was incorporated to match a stop on the fixed structure of the wing. The base and walls of this recess were increased in thickness to give the necessary strength and to match the geometry of the stop. The upper flange of the rib was designed to be on the underside of the skin for neatness but was later changed to the upper side to assist fabrication.

The trailing edge member was machined from a fiberglass laminate to the required wedge section. Recesses were machined for the upper and lower skins so that the edges of the boron would be protected. This feature also gave a better seal, helping to prevent the ingress of moisture. The width of this member was enough to ensure that the honeycomb core thickness was not reduced below one-eighth of an inch.

A landing gear fairing structure had to be attached to the underside of the flap. It was not within the scope of the program to redesign this fairing, so that the production part was used. Again, for interchangeability reasons it was decided to retain the existing eight attachment screws and some means of accepting these screws within the structure had to be devised. The fairing was dimpled to accommodate the countersunk heads of the screws and these dimples, in turn, required countersinks in the lower skin surface. This was accomplished by designing special inserts that passed through holes in both skins, Figure 12, being flush with the lower surface and projecting slightly through the upper. These inserts were in four pairs of differing lengths and were manufactured by Delron in accordance with their standard insert practice. Adequate locking of the threads was provided, and two holes in each insert enabled a filler resin to be injected into cavities in the core to hold the inserts in place. Organoceram resin was used as the filler and the same material was used as an adhesive around the edges of the core. It was later discovered that this filler resin and the adhesive accounted for 2.7 pounds of the flap weight, or 16 percent of the total. Clearly, here was a place for improvement in any future design.

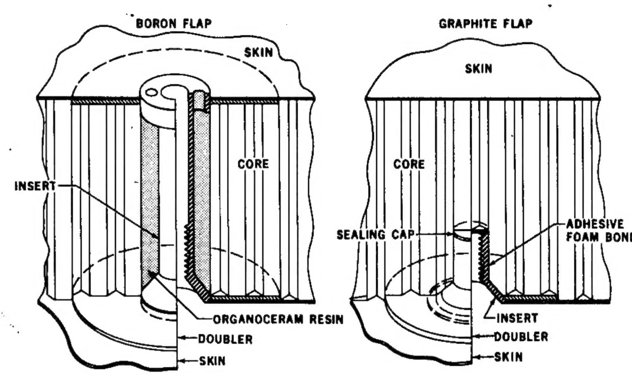

FIGURE 12. INSERT DESIGNS

Fabrication

The main innovation in the fabrication of the flap was the design and development of an automatic tape-layup machine, Figure 13. It is the purpose of this machine to lay a tape on a flat caul plate by traversing the head along the length of the bed. At the end of the required travel, the tape is automatically cut and the head is returned to begin a new stroke. Between strokes, the head is traversed by one tape width so that the tape layed on each stroke is adjacent to the one layed previously.

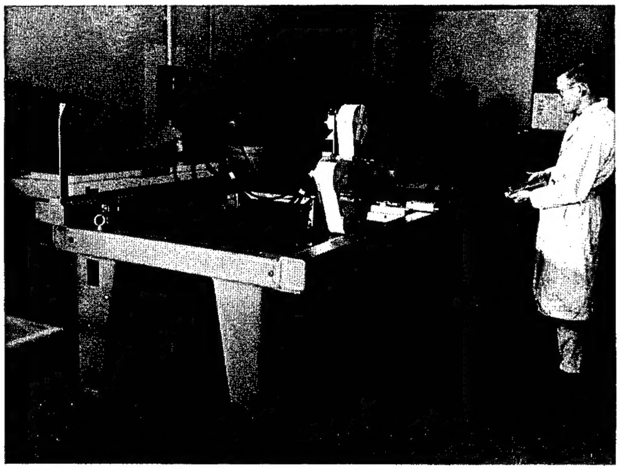

FIGURE 13. AUTOMATIC TAPE LAYUP MACHINE

The procedure is continued until the layer is complete and the succeeding layer is then commenced after the bed of the machine has been rotated to give the desired filament direction. Some difficulty was experienced with this machine, mainly associated with matching the edges of the tape. The tape has to be positioned accurately on the paper backing if the desired results are to be achieved. The problem was aggravated by the fact that as the tape was unwound from the drum, there was a tendency for the paper to separate from the tape and form wrinkles. This separation increased the misalignment problems of the tape. Despite these and other difficulties, satisfactory skins were produced on this machine and valuable lessons have been learned. A new tape laying head has been designed and is now being manufactured. This new head will be used for the fabrication of the graphite flap.

After the layup of the skin was completed the skin was cut roughly to shape, bagged, and placed in an autoclave to be cured under pressure and temperature. Final shaping was done with a diamond saw and the holes were cut with hollow diamond drills. The honeycomb core was shaped by standard production methods and this and the other detail parts presented few problems.

The leading edge hinge and its associated angle member were first bonded together as a subassembly. A further secondary bonding operation then joined the leading and trailing edges, the inboard rib fitting, and the fiberglass dish to the core. Finally the skins and doublers, the outboard closing rib and some sheet metal details were added in the primary bonding stage. The few remaining details were afterwards added with a room-temperature adhesive.

The primary bonding jig was arranged to incorporate the preload twist in the flap and the finished twist was close to that specified. However, there were dimensional changes along the leading-edge, where aluminum parts were bonded to the boron skins along a length of about 72 inches. The ±45° boron had a lower coefficient of thermal expansion and resulted in a stretching of the aluminum during the cooling period of the cure. There was also a slight mismatch of the piano hinge in relation to the portion of hinge on the rib fitting and the result was that every lobe of the piano hinge had to be machined away by a back-spotface cutter before the flap could be fitted onto the wing structure. The bonding jig, which had to be produced on a limited budget, did not adequately locate all of the parts and there was some movement of the loose doublers during the cure.

Inspection methods were visual, by X-ray on individual skins and by heat sensitive paint to show the quality of the bond to the skins. After weather-sealing had been applied, the final flap was immersed and checked for leaks. This flap was given a coat of white paint over its entire surface except for a chordwise strip that is left unpainted so that the quality of the skins can be inspected throughout the life of the flap.

Three flaps were made, one for fatigue testing, one for ultimate strength testing and the third as a fully flightworthy article. There was a gradual development in design and fabricating techniques as the lessons from each flap were learned. Full details of the fabricating processes are given in other papers.[3] [4]

Testing

The first two flaps were mounted, in turn, upside down in the test rig and the air load applied by means of compression pads loaded through a whiffle-tree linkage. A hydraulic jack was used to provide the actuator load. Strain gages were fitted to the skins and deflection readings were taken along the trailing edge with dial gages. The fatigue loads were represented by the summation of three different loading cycles. After surviving more than twice the aircraft life without damage, the flap was loaded in the primary flaps-down condition and failed at a load of 181 percent of limit load. Failure was initiated at the inboard forward corner of the flap and a fold ran at roughly 45° from this corner. Delamination of the skin from the core occurred very locally in the region of this fold, and some secondary delamination took place along the edges of the rib fitting and the fiberglass dish.

It should be noted that this secondary delamination was in a region where the bonding to the core was of poor quality. This was apparent after the inspection with heat sensitive paint and was attributed to a poor match between the fitting and core heights. Although the core was machined to fine limits, the rib fitting was a standard production part and not closely controlled. Consequently, there were places where the rib was too high to enable the full bonding pressure to be applied to the adjacent regions of the core.

In the ultimate strength test a load of 244 percent of limit load was reached before the hinge fitting on the testing rig broke. Since the strength of the flap had already been well demonstrated,

the testing was discontinued and the second flap was kept for exhibition purposes.

The third flap was proof tested and has since undergone thorough inspection, including leak testing by immersion. Before flight testing, it is proposed to submit it to vibratory tests to determine modes and frequencies.

Basic Graphite Properties

Graphite is in an earlier stage of development than boron and shows promise as a high strength structural material that is easy to work. At present it is about the same cost as boron but has a potential for becoming very much cheaper. It will also be attractive in applications where high temperatures are concerned. In selecting the monolayer values given in Table I, the properties of Thornel 50 material were used as a rough guide. Actually, there are a number of competing materials and these are being evaluated. Graphite is competitive with boron only if the specific modulus is critical, but the strength potential is higher than that indicated and a continual improvement may be expected. In the case of the flap, it is possible to use the same skin thickness as for boron and to take advantage of the reduced density.

In practice, there is a constraint on the choice of skin thickness imposed by the thickness of a single ply. For example, Thornel 50, which is the most readily available material, was found to have a cured ply thickness of 0.008. This narrowed the choice to 4-ply, which gave a heavier skin than on the boron flap, or 3-ply, which was slightly lighter. Again, the 3-ply skin was treated with suspicion because the imbalance in shear made analysis difficult. Furthermore, although thermally balanced about the mid-plane, there was a thermal out-of-balance in the plane of the skin that could produce a torsional twisting of the completed flap.

The initial selection of a suitable material is linked to the availability and delivery situation. The large ply thickness of Thornel 50 is due to the fact that each bundle of individual filaments is twisted into a yarn that cannot easily be compressed. Other filaments, notably the RAE type, are untwisted, and ply thicknesses as low as 0.002 can be achieved. In order to make early use of the Thornel 50, investigations are in progress to produce a thinner ply by placing the bundles of filament farther apart, so that the percentage graphite volume is unchanged. If this is successful, and if laminate properties are not adversely affected, a 4-ply skin of approximately 0.020 total thickness would result. One implication of this approach is that a glass scrim cloth would be required to hold the tape together, as was the case with boron. Later flaps in the program could take advantage of any change of material that is subsequently shown to be beneficial.

The Graphite Flap

In redesigning the flap to make use of the recently available graphite filamentary materials, the opportunity was taken to introduce a number of refinements. The graphite filaments were different from those of boron in several respects, the main factor affecting design being the extremely small diameters of the individual filaments. This eliminated the previous prohibition on small curvatures so that the design of the skin was no longer limited to flat panels. The skin is actually to be laid flat in one large piece and folded, while in its uncured state, about the trailing edge. Secondary folds are then made to form the outboard closing rib member, Figure 14. The recess for the stop, and skin reinforcing layers are all completed at this stage so that the complete skin is cured at one time, thus eliminating all loose doublers. A male forming tool assists the fabrication of the skin and remains in position during the cure.

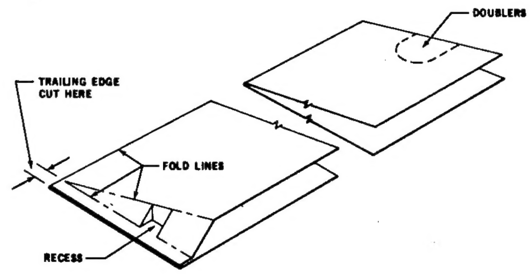

FIGURE 14. GRAPHITE FLAP SKIN ONE-PIECE CURED LAMINATE

The cured skins are sprung apart so that the shaped core can be inserted. A lighter core is used than for the boron flap, a quarter-inch cell size being selected. The resulting density is 2.3 pcf, which is about the lightest core that can withstand the bonding pressure. The leading edge of the core is bounded by a molded graphite laminate on which is bonded portions of the production metal hinge, Figure 15. The hinge is cut into short lengths so that the accumulation of dimensional errors will be avoided by locating each piece individually in the bonding jig. Initially, it was intended to make a molded graphite leading edge with integral hinge lugs but there was some doubt about the wear properties of the graphite hole under repeated loads. At the inboard end of the flap, the previous rib fitting, the fiberglass dish and the associated sheet metal parts are replaced by a molded graphite fitting, which is described in detail later.

The access door and landing gear fairing are again retained as production metal parts although a feasibility study has been made on a fairing made of graphite. The inserts, into which the fairing

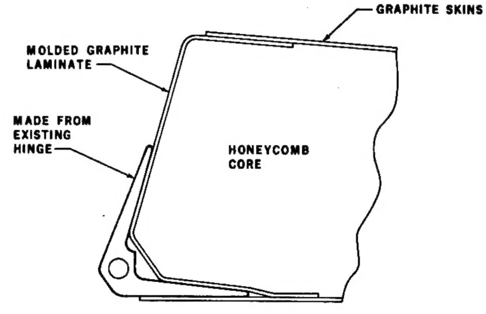

FIGURE 15. GRAPHITE FLAP LEADING EDGE

attachment bolts are screwed, have been redesigned to save weight and cost by fixing them to the lower skin only and by making them all the same length, Figure 12. The aftermost pair of inserts are restricted in available depth, and in order to incorporate adequate sealing, the bolt length has been reduced slightly. The organoceram filler, that was previously so costly in weight, is being replaced by adhesive foam, and this material is also used to stiffen the extreme trailing edge beyond the honeycomb. A comparison of numbers of parts and of flap weights is given in Table II.

TABLE II
COMPARISON OF FLAP DESIGNS

	METAL	BORON	GRAPHITE
NUMBER OF PARTS:			
COMPONENTS	65	63	34
ATTACHMENTS	215	21	13
TOTAL	280	84	47
WEIGHT: LB/FLAP			
PARTS	21.6	14.2	11.0
ADHESIVE	–	1.4	1.2
FILLER	–	1.3	1.0
TOTAL	21.6	16.9	13.2

The progressive reduction in numbers of parts is significant because if the fabrication is simplified enough it could offset the initial high cost of the filamentary materials. It should, of course, be noted that the lower cost glass filaments could be used to advantage in applications where strength, rather than stiffness, is critical. In the flap programs, the selected materials are used in order to gain experience with them, even though, in some instances, they are not the best choice for the particular application. Compared with the metal production flap, the weight saving for boron and graphite is 22 and 39 percent, respectively.

Molded Graphite Rib Fitting

The rib fitting was thought to be a suitable component for the investigation of a graphite molding process, Figure 16. Geometrically, it is constrained by the actuator lugs, the leading edge hinge, the flap profile and by the clearance requirements of the actuator linkage. By replacing the channel section of the aluminum fitting by a zee section, a considerable weight saving can be effected. The lower flange helps to support the actuator lugs, the hinge, and the lower skin, thus removing the need for the previous array of reinforcing members. The double lug for the actuator attachment is integral on one side with the web of the fitting. The other side, being more flexible, is encouraged to take more of its share of the load. This is done by leaving some material to bridge between the two lugs, thus providing additional bending support. The holes in the lugs are reinforced with metal bushings for wear, rather than for strength.

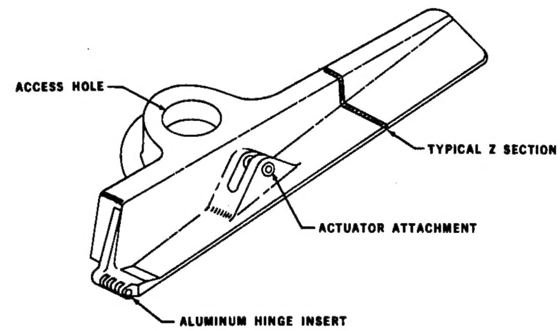

FIGURE 16. MOLDED GRAPHITE RIB FITTING

In the case of the lugs at the hinge end, the geometry is so constricting that it is not possible to use metal bushings. A complete metal part incorporates the lugs and transfers the loads into the molding. A metal blade is the main load path into the web of the fitting and bending stiffness is provided to take care of the offsets of the load. The metal part is molded into the fitting during the cure and the resulting bond has to have adequate strength to transfer the loads.

Between the actuator lugs and the hinge there is high shear in the web of the fitting and layers of tape are placed at ±45° to take the load. These layers have to be carried around the corner and joined to the cap material. Bending loads peak at the actuator lug position and sufficient zero degree layers are added to provide the required strength at any station. Layers in other directions are added where needed and chopped filament material is used where the strength is not so critical. At all stations the layup pattern has to be arranged so that thermal balance is achieved.

Structurally, it is desirable to make the cut-out reinforcing structure an integral part of the fitting, as shown in Figure 16. However, this involves an increase in complexity of the mold tool that may not be justifiable at this stage. A separate dish molding, as before, would be an easier first step, to be followed later by the one-piece fitting. The dish is a double curvature part for which chopped filaments would be ideal, since strength requirements are not high. An interesting alternative proposal is to use glass in this area, which would result, for the one-piece design, in a fitting in which the materials are mixed within the molding.

In designing the lug-ends in graphite, and for joints generally, the results of the Douglas Joints and Cutouts Program, now being undertaken for the Air Force, have been used.[5]

Concluding Remarks

It is now well established that satisfactory components can be made with boron materials and that considerable weight savings can be achieved. Graphite offers the same advantages with the prospect of much improvement in the future. The development of high temperature resins and other types of matrix materials will soon widen the field

of applicability. The high cost of such materials at present is inhibiting their widespread acceptance, but there are a number of special instances where cost-effective parts can be made. A cost and strength trend curve for boron is shown in Figure 17, to indicate the improvements that are likely to be achieved in the future. In the commercial field it may be 10 years or more before the confidence level is high enough to warrant extensive use of these materials. However, the gradual introduction of these and related materials would appear to be inevitable if the present scale of research is maintained.

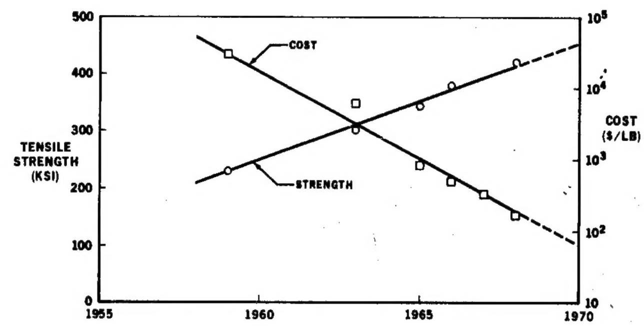

FIGURE 17. COST AND STRENGTH TRENDS, BORON FILAMENTS

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